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RESEARCH MEMORANDUM

AERODYNAMIC STUDY OF A WING-FUSELAGE COMBINATION EMPLOYING A WING SWEPT BACK 63°.- CHARACTERISTICS FOR SYMMETRICAL WING SECTIONS AT HIGH SUBSONIC AND MODERATE SUPERSONIC MACH NUMBERS

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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AERODYNAMIC STUDY OF A WING-FUSELAGE COMBINATION EMPLOYING A WING SWEPT BACK 63°.– CHARACTERISTICS FOR SYMMETRICAL WING SECTIONS AT HIGH SUBSONIC AND MODERATE SUPERSONIC MACH NUMBERS

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SUMMARY

Results of wind-tunnel tests are presented for a wing with the leading edge swept back 63° and of symmetrical section in combination with a body at Mach numbers from 0.5 to 0.95 and from 1.09 to 1.51. The test Reynolds numbers varied from 0.35 to 0.52 million. Measured lift, drag, and pitching-moment coefficients for the configuration are compared with corresponding calculated characteristics. The results indicate that available analytical methods may be used with confidence in the prediction of the variations with Mach number of the lift of highly swept wings. It is also found that the measured trends of the minimum drag coefficient with Mach number compare favorably with those indicated by theory throughout the Mach number range of the tests. The low Reynolds numbers of the tests virtually invalidate any quantitative comparison of the measured characteristics of pitching moment and drag due to lift with those calculated by the methods of inviscid theory. However, the results are useful in indicating gross changes with Mach number of the aerodynamic-center location and the approximate magnitude of the maximum lift-drag ratio to be expected for a highly swept wing configuration at moderate supersonic Mach numbers.

INTRODUCTION

R. T. Jones has indicated in reference 1 the possibility of developing practicable values of maximum lift-drag ratio at supersonic Mach numbers with wings swept well behind the Mach cones emanating from the leading edges. To examine this possibility experimentally and to determine the aerodynamic properties of such highly swept wings under other flight conditions, an extensive wind-tunnel investigation has been undertaken in several facilities of the Ames Aeronautical Laboratory. Tests have been completed (references 2, 3,

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and 4) and others are in progress to determine the principal aerodynamic characteristics of a configuration suggested by the analysis of reference 1 over a broad range of Mach and Reynolds numbers. The present investigation was initiated in the Ames 1- by 3-1/2-foot high-speed wind tunnel to determine the respective variations with Mach number of the lift, drag, and pitching-moment coefficients of the selected configuration at transonic Mach numbers beyond the reach of other currently available wind tunnels.

The configuration consists of a wing with the leading edge swept back 63° in combination with a body designed to have the minimum drag at supersonic speeds for a given length and volume. The wing was designed from aerodynamic considerations and from the structural criterion of reference 1 to provide useful maximum lift-drag ratios at Mach numbers up to about 1.5.

The results of the investigation are of additional value as an indication of the applicability of a number of linearized theories at Mach numbers for which they have been considered invalid.

SYMBOLS

A	aspect ratio $\left(\frac{b^2}{S}\right)$
b	wing span, feet
c	local chord, feet
$\bar{c}$	mean aerodynamic chord $\left(\frac{\int_0^{b/2} c^2\,dy}{\int_0^{b/2} c\,dy}\right)$, feet
C_D	drag coefficient $\left(\frac{\text{drag}}{qS}\right)$
$C_{D_{min}}$	minimum drag coefficient
ΔC_D	increment in drag coefficient $(C_D - C_{D_{min}})$
C_L	lift coefficient $\left(\frac{\text{lift}}{qS}\right)$
$C_{L_{C_{D_{min}}}}$	lift coefficient corresponding to minimum drag coefficient

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ΔC_L	increment in lift coefficient $\left(C_L - C_{L_{C_{D_{min}}}}\right)$
$\frac{\Delta C_D}{\Delta {C_L}^2}$	drag-rise factor
C_m	pitching-moment coefficient $\left[\frac{\text{moment about } (\bar{c}/4)}{qS\bar{c}}\right]$
$(L/D)_{max}$	maximum lift-drag ratio
M	free-stream Mach number
q	free-stream dynamic pressure, pounds per square foot
R	Reynolds number, based on mean aerodynamic chord
S	wing area, square feet
y	lateral coordinate measured from the plane of symmetry, feet
α	angle of attack, degrees
$\Delta\alpha$	jet boundary correction to angle of attack, degrees
λ	taper ratio $\left(\frac{\text{tip chord}}{\text{root chord}}\right)$

APPARATUS AND TEST METHODS

The tests were conducted in the Ames 1- by 3-1/2-foot high-speed wind tunnel which is equipped with a flexible nozzle permitting tests at both subsonic and supersonic Mach numbers.

The model, which was constructed of steel, was of the same basic configuration as that of the tests of reference 2. The wing consisted of NACA 64A-006 sections in the streamwise direction. Major dimensions of the model and the meridian curve of the body are shown in figure 1. The model was supported from the rear of the body by a sting that was shielded from direct air loads. (See fig. 2.)

Lift, drag, and pitching moment were measured on a three-component strain-gage balance at angles of attack varied in approximately 1° increments from -2° to 7° and at Mach numbers from 0.5 to 0.95 and from 1.09 to 1.51. The Reynolds number varied from 0.35 to 0.52 million as shown in figure 3.

Schlieren observations of the flow field about the model were made at supersonic Mach numbers. Several representative photographs are presented in figure 4.

REDUCTION OF DATA

All forces and moments were measured about the wind axes and are presented in the conventional coefficient form. At subsonic Mach numbers the following jet boundary corrections to the angle of attack and drag due to lift, determined by the methods of reference 5, were applied to the data:

$$\Delta\alpha = 0.398\ C_L$$

$$\Delta C_D = 0.007\ C_L{}^2$$

Blockage corrections were found to be negligible for the model investigated and were not applied to the data.

Possible interference effects between the support system and the model were eliminated by correcting the measured drag for the force resulting from the difference between the pressure measured at the base of the body and the free-stream static pressure. By this means, which was also employed in reference 2, the base drag of the body is subtracted from the total drag of the model. The measured drag values were further corrected for the effects on the body of the static pressure gradients of the free stream.

Although zero lift at zero angle of attack was obtained at all subsonic Mach numbers, where the stream inclination is known to be negligible, this was not true at several supersonic Mach numbers. In these instances, the angles of attack were corrected by the amount required to shift the angle of zero lift to the origin. The drag coefficients were correspondingly corrected for the corrections to the angles of attack. The stream-angle correction did not exceed 1°.

RESULTS AND DISCUSSION

Lift

The curves of lift coefficient as a function of angle of attack for the model at all test Mach numbers are presented in figure 5. These have been drawn as straight lines although the test points at the lower Mach numbers indicate actual variations that are somewhat nonlinear. At the low test Reynolds numbers the nonlinear characteristics could be caused by large differences in the thicknesses of the boundary layers or in the respective extent of the separated flow regions on the upper and lower surfaces of the wing at the trailing edges. In either case the camber of the wing sections would be effectively altered.

In figure 6, the variation of the mean lift-curve slope with Mach number for the model is compared with the theoretical variation for the wing and with the results of the tests of references 2, 3, and 4. The calculated values were obtained by the methods of references 6 and 7.

At subsonic Mach numbers it is seen that no lift divergence occurs and that good agreement exists with the type of variation of lift-curve slope with Mach number predicted by the use of lifting-line theory and the extensions of the Prandtl-Glauert rule described in reference 6. The agreement of the present results with those of reference 3, at a much greater Reynolds number, is also good. The increment of lift-curve slope contributed by the body is appreciable, as is indicated in figure 6 by the results of reference 4. This body lift accounts for a major portion of the difference between the present results and the calculated results for the wing alone.

At supersonic Mach numbers the agreement between the results obtained with the model and the calculated values varies from good to fair with increasing Mach number. It should be noted that the effect of the body has not been considered in the calculations. At 1.5 Mach number, agreement of the present result with that of reference 2[1] is excellent, but both results are somewhat smaller than the value calculated at that Mach number. As was pointed out in reference 2 the lack of agreement with inviscid theory is associated with the extensive laminar separation existing over the aft sections of the upper surface of the wing at moderate angles of attack. This flow separation results in an effective change in the airfoil camber that decreases the lift.

Drag

The variations of drag coefficient with lift coefficient of the model at the several test Mach numbers are presented in figure 7. In order to facilitate a study and comparison of experimental and calculated drag characteristics, it is convenient to separate the total drag into two components: minimum drag and drag due to lift.

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[1]The Reynolds number of 0.69×10^6 indicated on the figure for the results of reference 2 is based upon the mean aerodynamic chord of the model and corresponds to the value of 0.62×10^6 as used in the reference report based upon the mean geometric chord.

Minimum drag coefficient.- The variation of the minimum drag coefficient of the model with Mach number is shown in figure 8. It can be seen that no appreciable change in minimum drag coefficient occurred at subsonic Mach numbers although there was a slight rise in the values between 0.90 and 0.95 Mach number. This variation is also indicated by the results of reference 3 obtained at a higher Reynolds number. Values of minimum drag coefficient calculated by the methods of reference 8 are shown for both the fully laminar and fully turbulent boundary-layer conditions, since it was not possible to assess the exact proportions of each type of flow that existed on the model. It is seen that the observed and calculated trends of the minimum drag coefficient agree well at subsonic Mach numbers and the measured values fall well within the indicated skin friction limits. The observed increase in minimum drag coefficient occurring when the Mach number is increased to supersonic values is in good agreement with the predicted increase, and the measured values remain within the boundaries of laminar and turbulent skin friction. The calculated variation of the pressure drag component of the minimum drag coefficient of the wing at supersonic Mach numbers is based upon the results of reference 9, which apply specifically to symmetrical double-wedge sections. Justification for the application of the results of reference 9 to the rounded leading-edge profile of the present model may be found in reference 2. The correspondence of the present results with those of reference 2 at 1.53 Mach number and a somewhat higher Reynolds number is fair.

Drag due to lift.- The component of the drag coefficient that is due to lift is related to the maximum lift-drag ratio in the following manner for a wing of symmetrical section:

$$\left(\frac{L}{D}\right)_{max} = \frac{1}{2}\sqrt{\frac{1}{C_{D_{min}} \times \frac{\Delta C_D}{\Delta C_L{}^2}}}$$

where it is convenient to consider the drag due to lift in the form of $\Delta C_D/\Delta C_L{}^2$, termed the "drag-rise factor." Thus, it can be seen that the drag-rise factor can influence the maximum lift-drag ratio to the same degree as the minimum drag coefficient. The variation of the measured drag-rise factor is presented in figure 9 with two calculated variations that describe the limiting values of the drag-rise factor at each Mach number. The lower calculated curve is the variation of the minimum values of the drag due to lift, that is, the condition of complete realization of the theoretically available leading-edge thrust. At subsonic Mach numbers, this curve was obtained from reference 6 and the values are very nearly equal to $1/\pi A$, the value for a wing with an elliptic span loading. At supersonic Mach numbers above 1.43, the calculated optimum drag-rise factor was obtained by the methods of reference 7. The variation of the values between Mach numbers of 1.43 and 1.0 has been represented by a straight line. The use of $1/\pi A$ as the value of the drag-rise factor at a Mach number of unity can be justified by the analysis of reference 1. The upper curve represents the drag-rise factor for the case of zero leading-edge thrust at each Mach number. Since this case corresponds to the condition for which the resultant force acts normal to the chord line, the curve has been determined as the variation of the reciprocal of the experimental lift-curve slope. At subsonic Mach numbers, it is seen from the results shown in figure 9 that the available leading-edge thrust was not completely realized on the model. It is believed that this loss of leading-edge thrust was caused by flow separation near the leading edges that occurred at the low test Reynolds numbers. It is also noted that the measured drag-rise-factor variation virtually parallels the upper curve up to Mach numbers of about 1.2, but at the higher supersonic Mach numbers the experimental results approach the calculated lower limiting curve.

The cause of the discrepancy between the result of this report at the highest Mach number and that of reference 2 is not known.

Maximum Lift-Drag Ratio

In figure 10, the measured variation of the maximum lift-drag ratio with Mach number is compared with the experimental results of references 2 and 3 and the calculated variations. The latter variations are based upon the calculated minimum drag coefficients shown in figure 8 and the calculated optimum drag-rise factors shown in figure 9. By reference to figure 8, where the measured minimum drag coefficients are seen to fall well within the calculated limits, it is deduced that a major portion of the difference between the present measured and calculated maximum lift-drag ratios is due to the high values of drag due to lift observed at the low test Reynolds numbers. Furthermore, about 60 percent of the difference between the present maximum lift-drag ratios and those of reference 3 can be traced to the improved drag-rise factors accompanying the higher Reynolds number of the latter investigation. The closer agreement of the present results with calculated values at the higher supersonic Mach numbers is primarily a reflection of the corresponding trend of agreement shown by the measured drag-rise factor. The lack of agreement between the present result and that of reference 2 largely results from the discrepancy between the respective drag-rise factors noted previously.

The principal conclusion to be drawn from these results is that, for the type of plan form investigated, maximum lift-drag ratios measured at low Reynolds numbers cannot be assumed to be reliable for prediction of performance at the full-scale Reynolds numbers. It may also be concluded for the present case that the failure to realize theoretical maximum lift-drag ratios is due, in a large part, to the corresponding failure to realize theoretical drag-rise factors.

Pitching Moment

Curves of pitching-moment coefficient as a function of lift coefficient for each test Mach number are shown in figure 11. The results of reference 3 for the higher Reynolds number and the results of reference 2 are also plotted in the figure. The large differences between the present results and those of reference 3 that are apparent in figure 11 are believed to be due to the dissimilarity of the boundary-layer conditions existing at the widely different test Reynolds numbers. The correspondence of the present result and that of reference 2, at a Mach number of about 1.5 and at similar Reynolds numbers, is good.

The locations of the aerodynamic center of the model at the various Mach numbers have been determined from the slopes of the pitching-moment curves between 0 and 0.2 lift coefficient and are shown in figure 12.

Corresponding values from reference 3 are also shown as are values calculated by the methods of references 6 and 7 wherever applicable.[2] At subsonic Mach numbers, the present variation of location of the aerodynamic center with Mach number is similar to that of reference 3, although the rate of rearward movement at the higher Mach numbers exhibited by the present results is probably too large. Although the present results are not believed to offer a quantitative representation of aerodynamic-center locations at high Reynolds numbers, they roughly verify the magnitude of the total theoretical rearward shift of that parameter over the range of the test Mach numbers.

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[2] The method of reference 7 cannot be used at Mach numbers below that at which the Mach cones emanating from the trailing-edge apex crosses the leading edges (1.43 Mach number).

CONCLUSIONS

From the results of tests performed on a wing with the leading edge swept back 63° and of symmetrical section in combination with a body at Mach numbers from 0.5 to 0.95 and from 1.09 to 1.51, it is concluded that for highly swept thin wings of moderate aspect ratio:

1. The variations of lift and drag coefficients with Mach number are continuous and small.

2. The total rearward shift of the location of the aerodynamic center occurring between Mach numbers of 0.5 and 1.5 corresponds approximately to that predicted by the use of theoretical methods.

3. The lift characteristics can be estimated with reasonable accuracy by analytical methods for Mach numbers as high as 1.5.

4. The trend with Mach number of the values of minimum drag coefficient is similar to that indicated by theoretical methods.

5. Measurements of drag due to lift and pitching moment at subsonic Mach numbers and low Reynolds numbers cannot be considered quantitatively representative of the corresponding characteristics at much higher Reynolds numbers.

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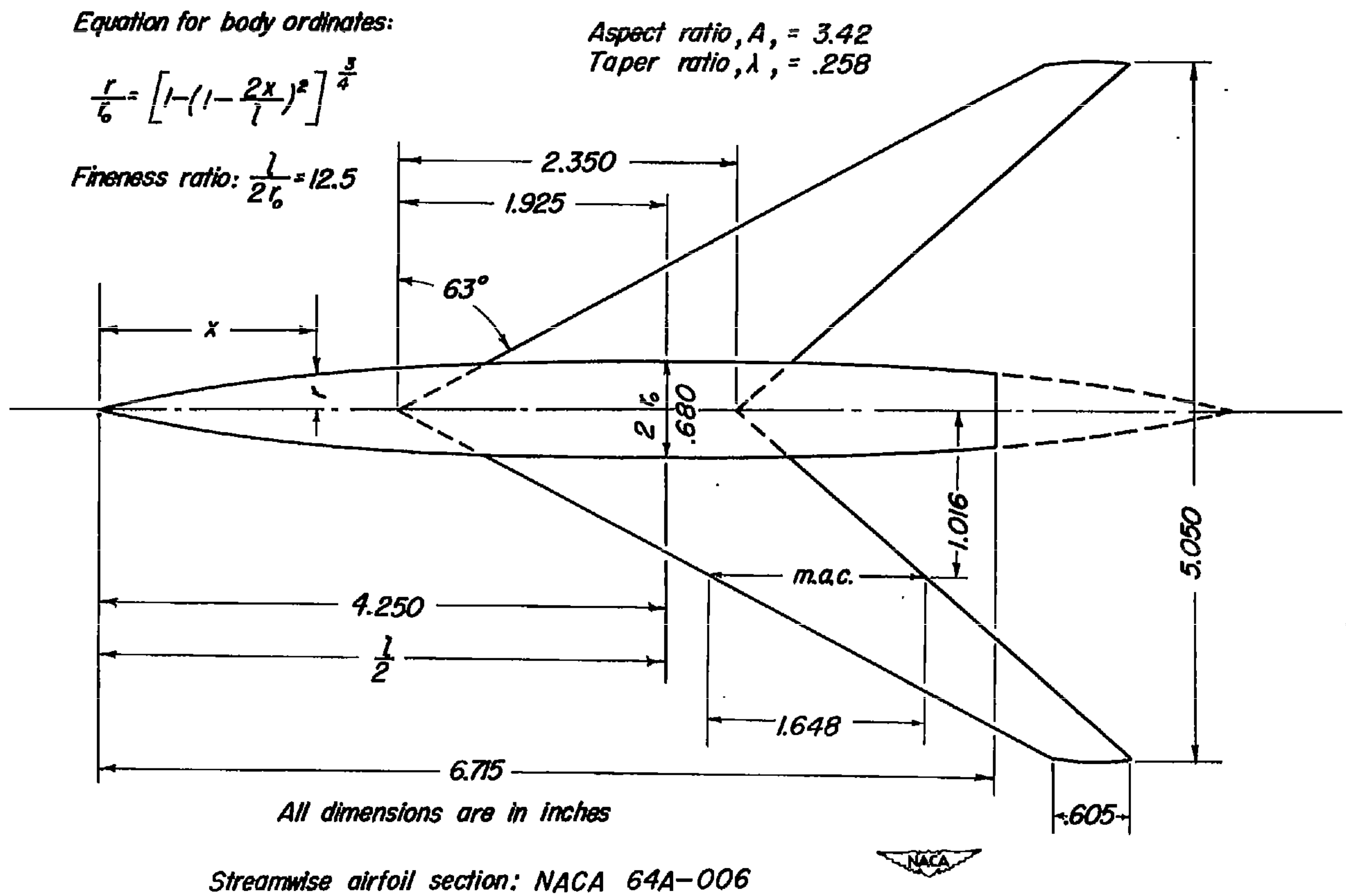
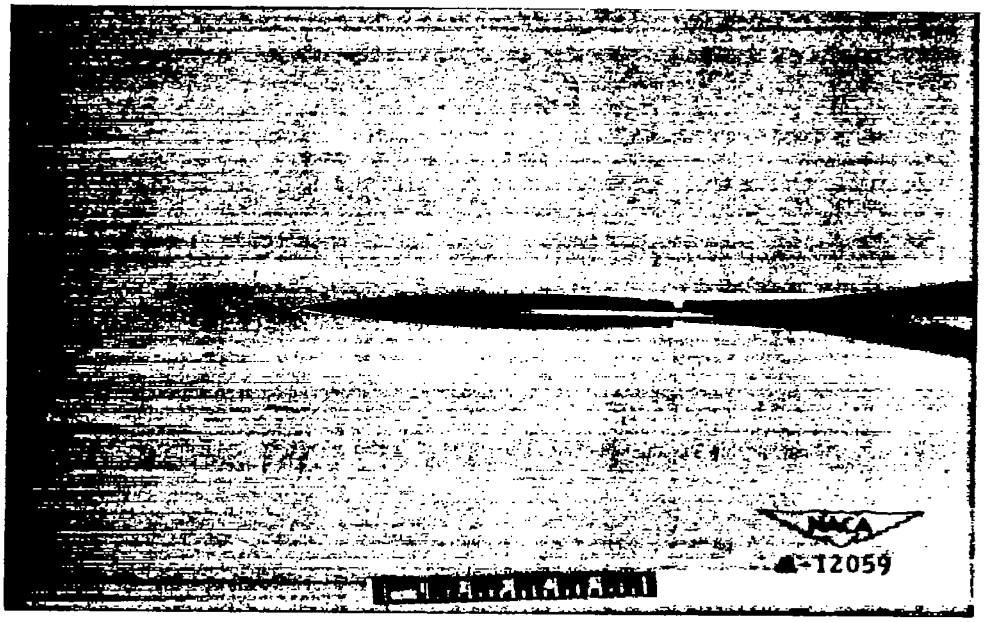
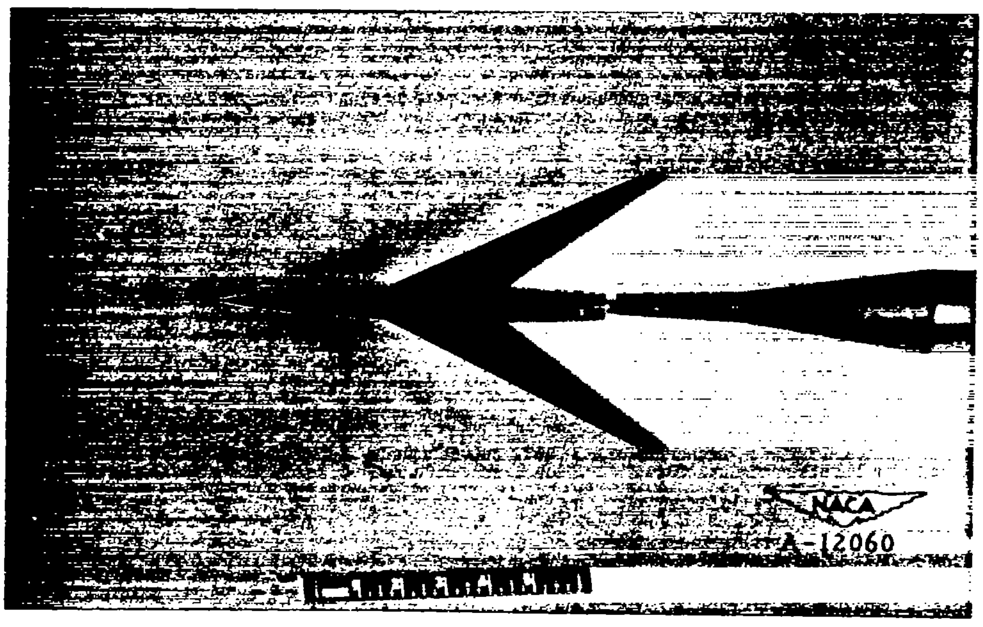


Figure 1.- Sketch of model illustrating principal dimensions of the wing and body.



(a) Side view.



(b) Plan view.

Figure 2.— Model on sting support.

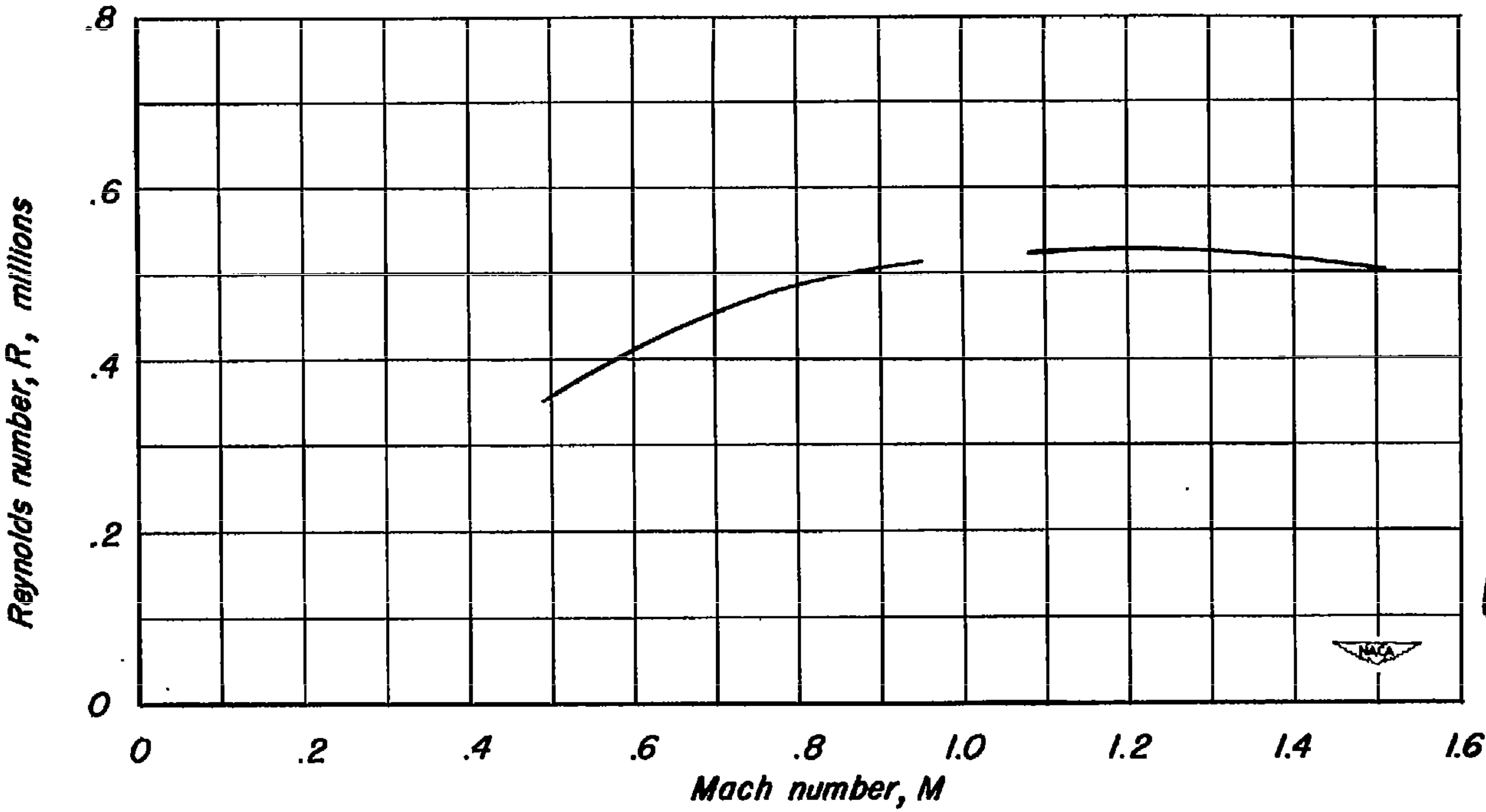
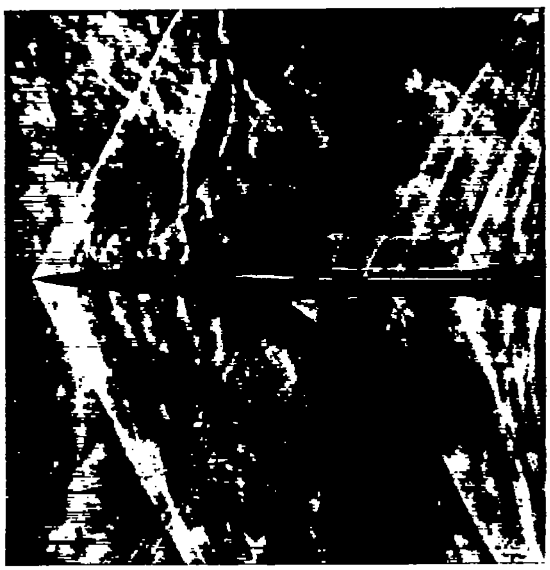


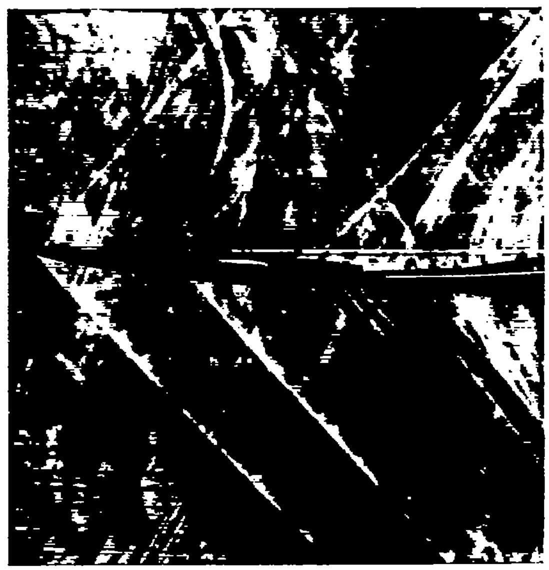
Figure 3.– Variation with Mach number of Reynolds number based on the mean aerodynamic chord of the wing.

(a) M = 0.

(b) M = 1.09, side view.



(c) M = 1.14, plan view.



(d) M = 1.14, side view.

Figure 4.– Typical schlieren photographs of the flow about the model at supersonic Mach numbers.

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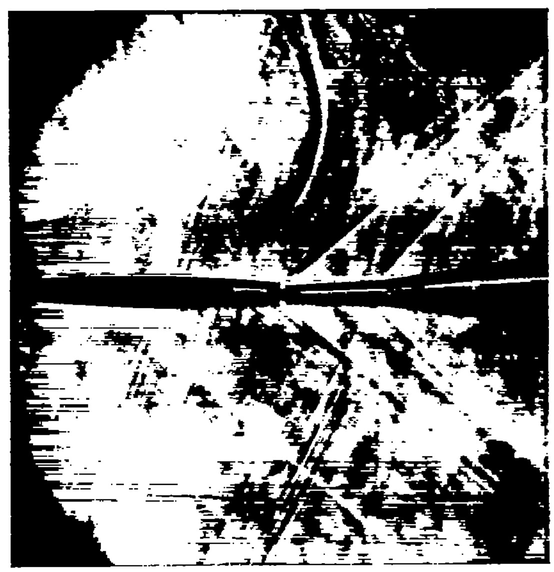
(e) M = 1.24, plan view.

(f) M = 1.24, side view.

(g) M = 1.51, plan view.



(h) M = 1.51, side view.

Figure 4.– Concluded.

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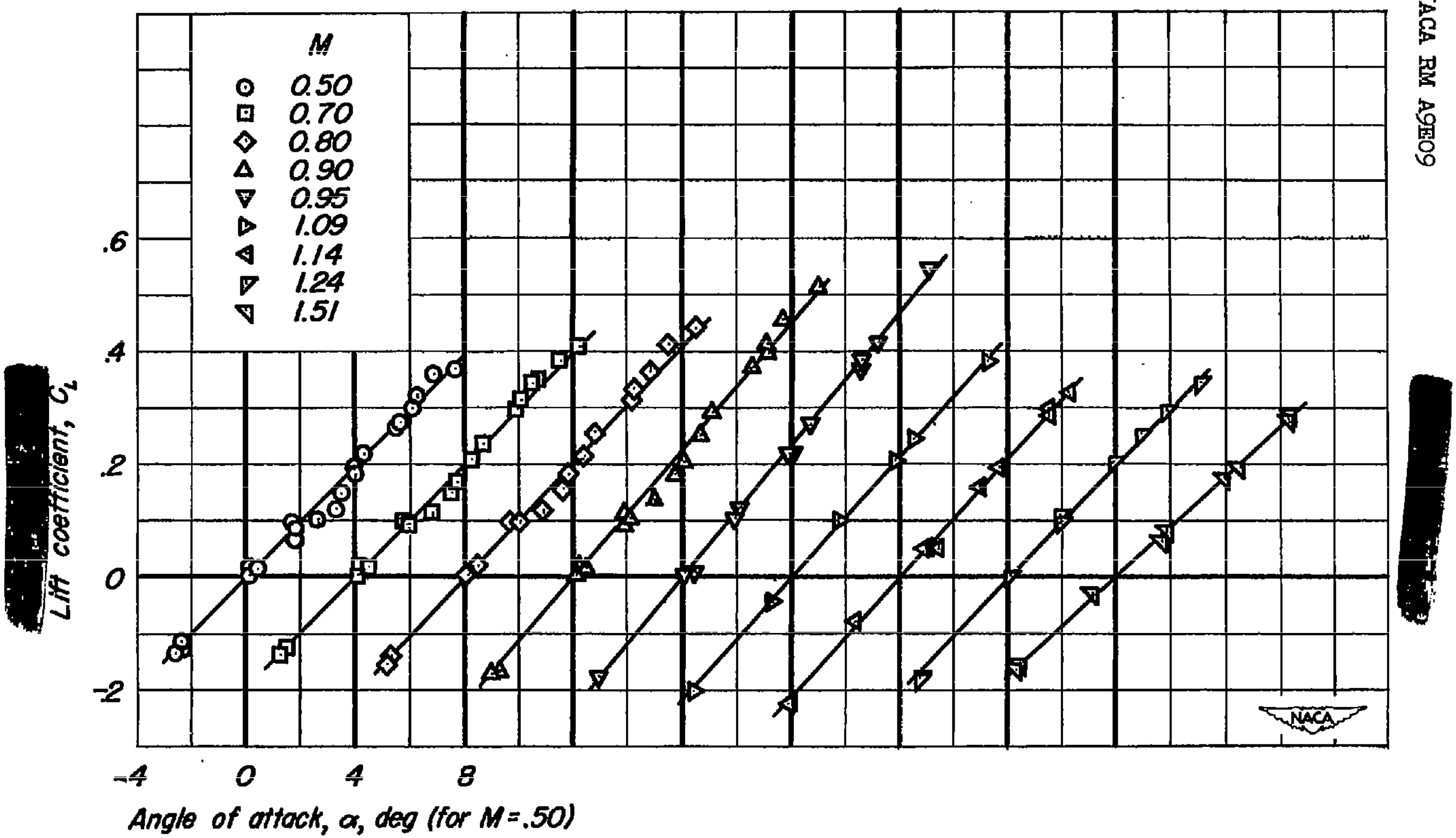

Figure 5.- Variations of lift coefficient with angle of attack at the various test Mach numbers.

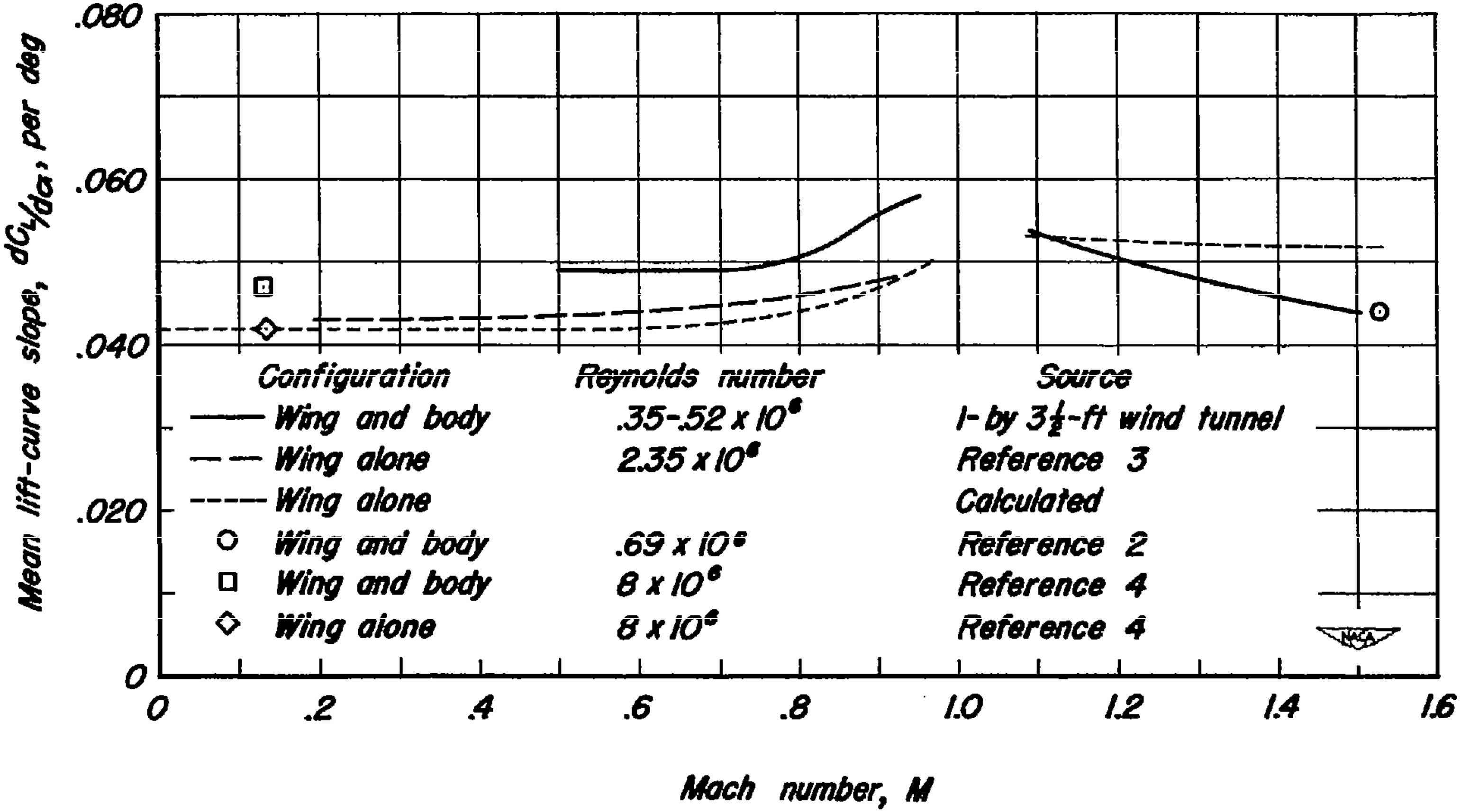

Figure 6.– Effect of Mach number on the mean lift-curve slope.

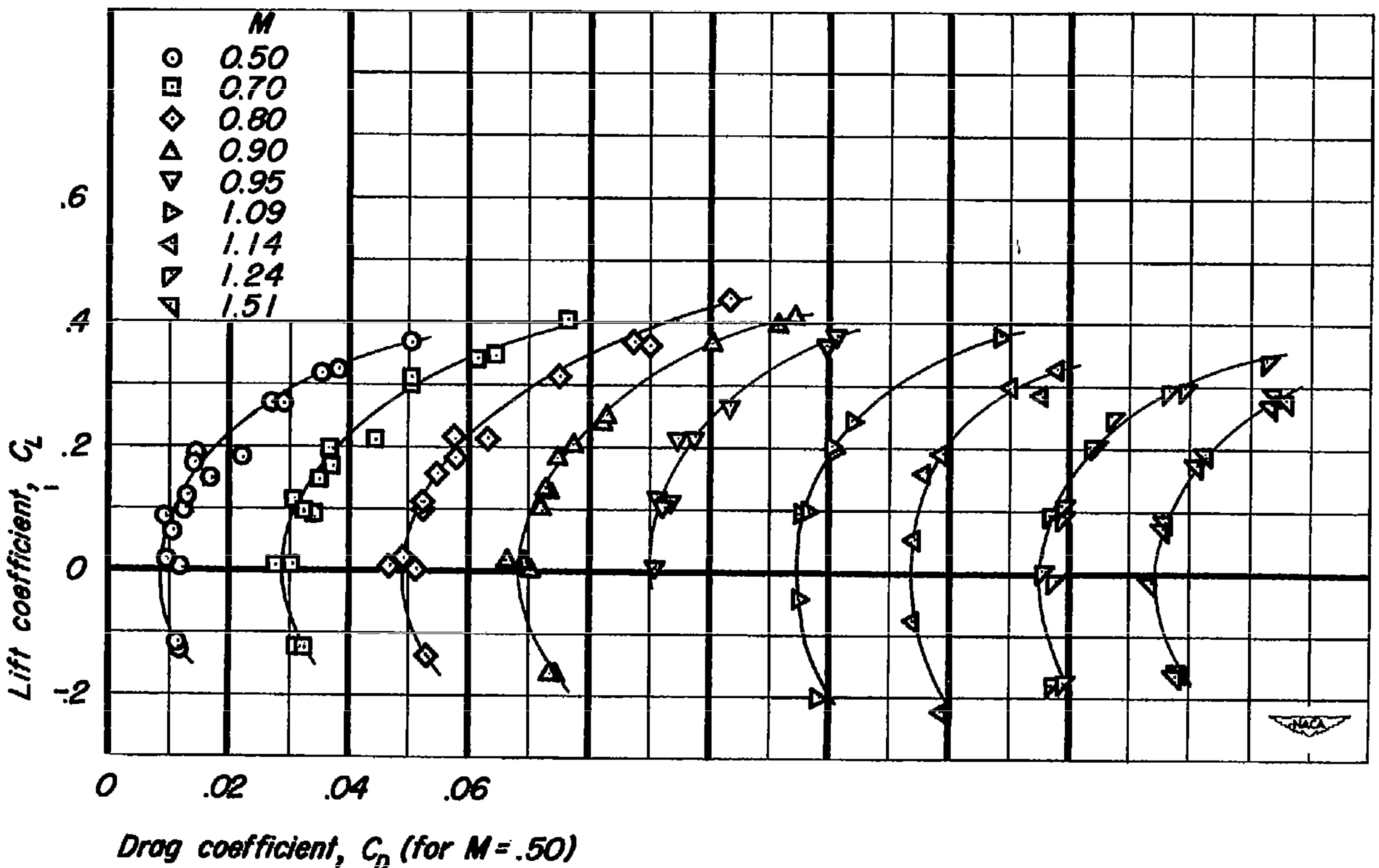

Figure 7.– Variations of drag coefficient with lift coefficient at the various test Mach numbers.

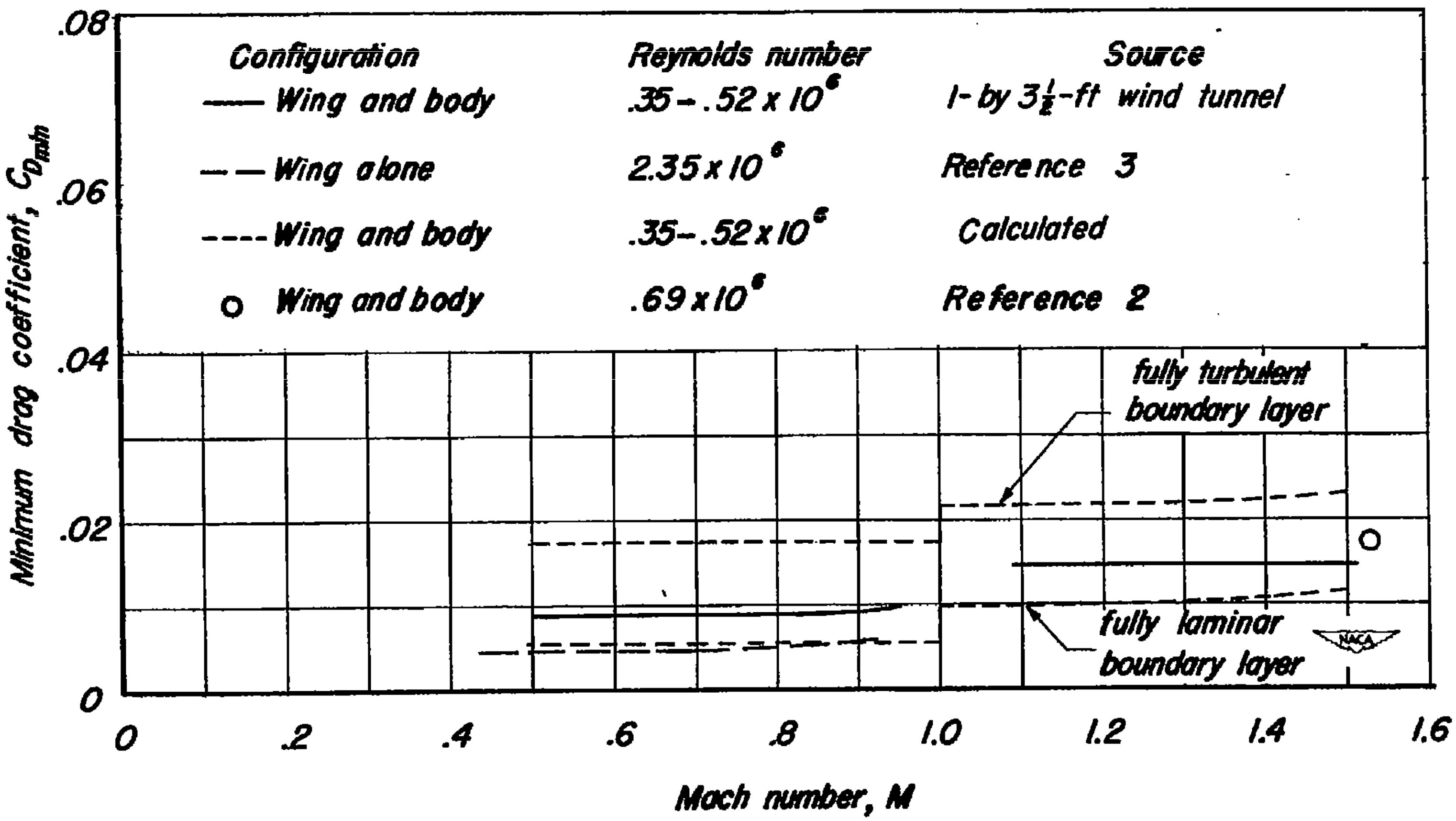

Figure 8.- Effect of Mach number on the minimum drag coefficient.

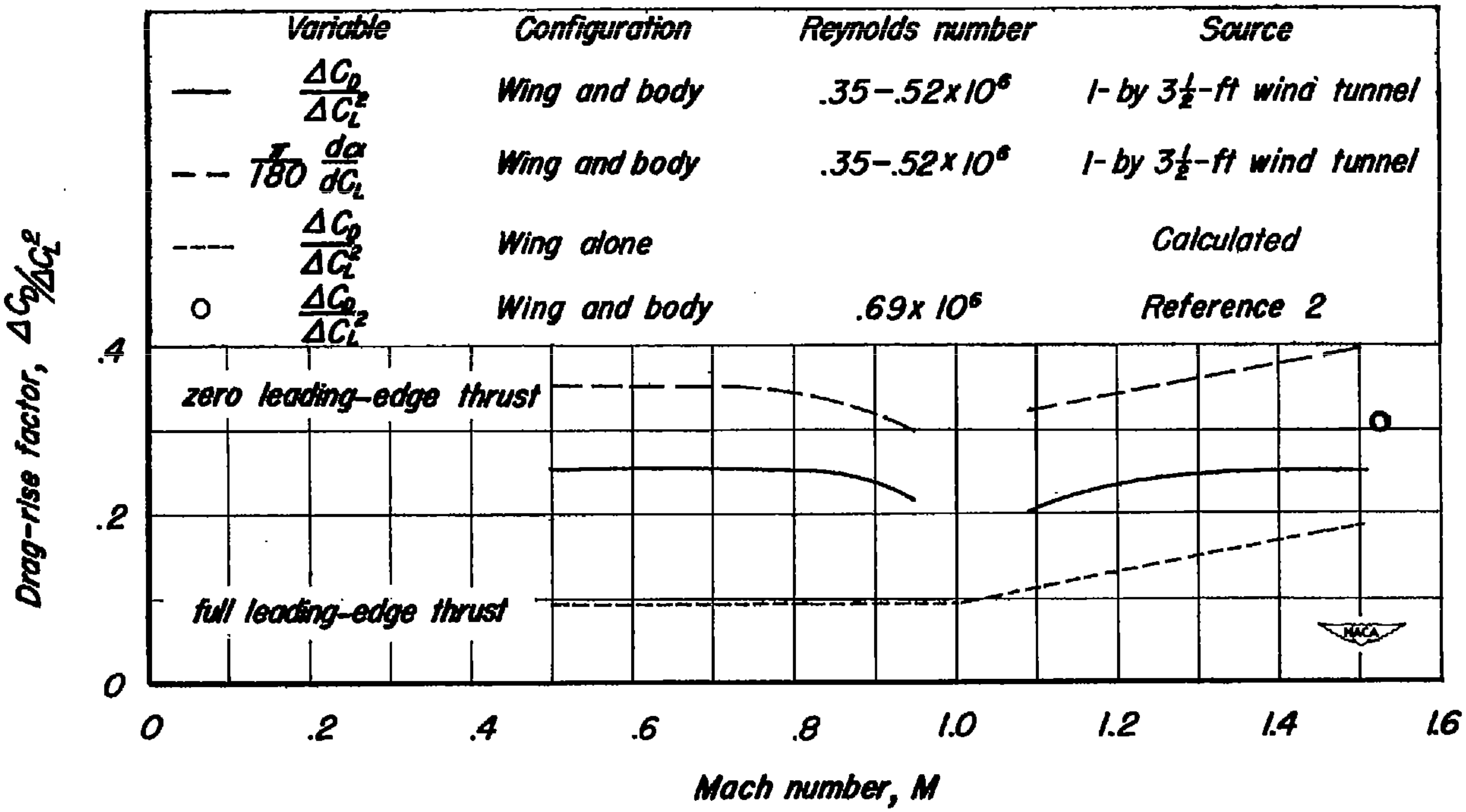

Figure 9.- Effect of Mach number on the drag-rise factor.

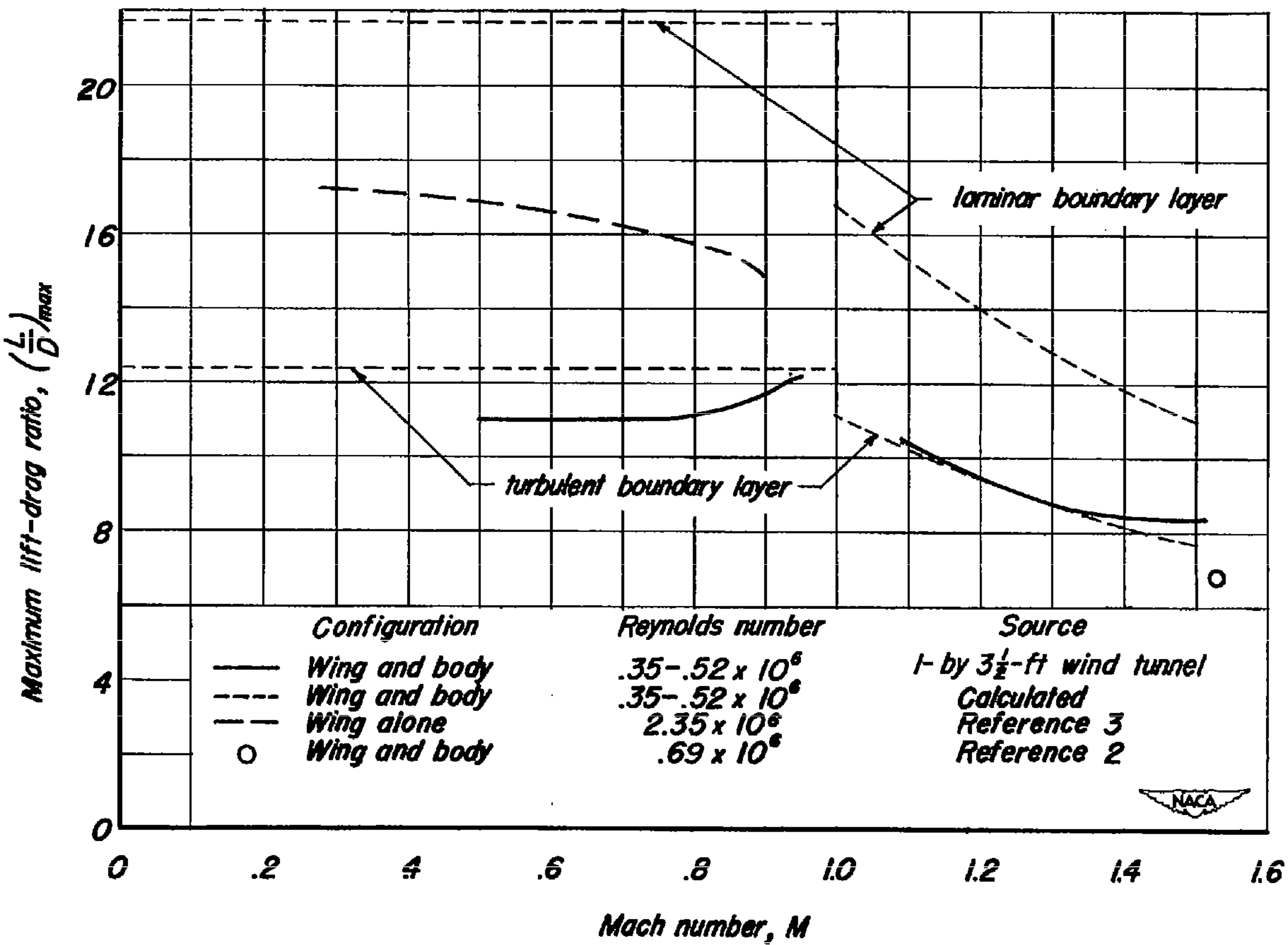

Figure 10.– Effect of Mach number on the maximum lift-drag ratio.

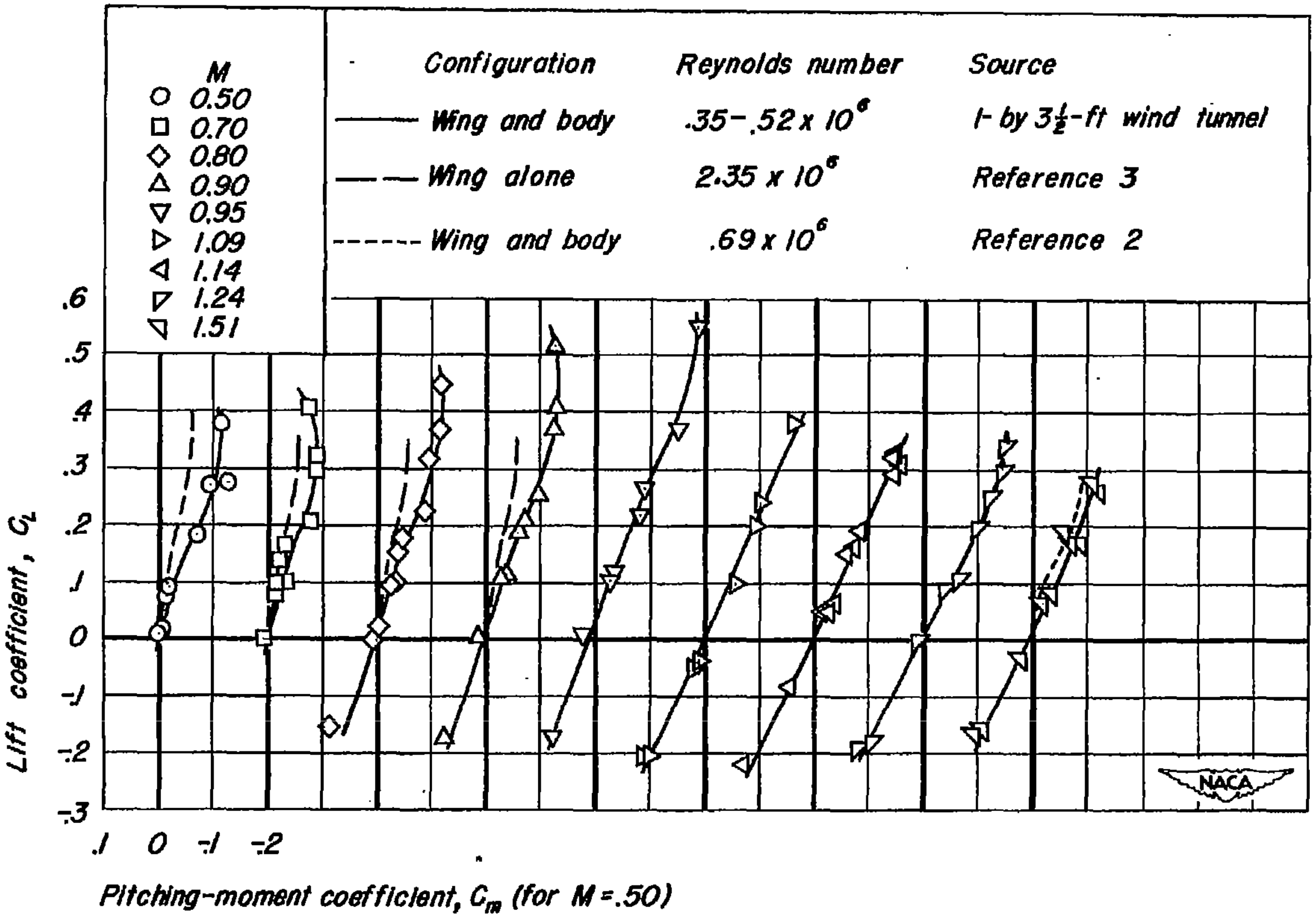

Figure 11.- Variations of pitching-moment coefficient with lift coefficient at the various test Mach numbers.

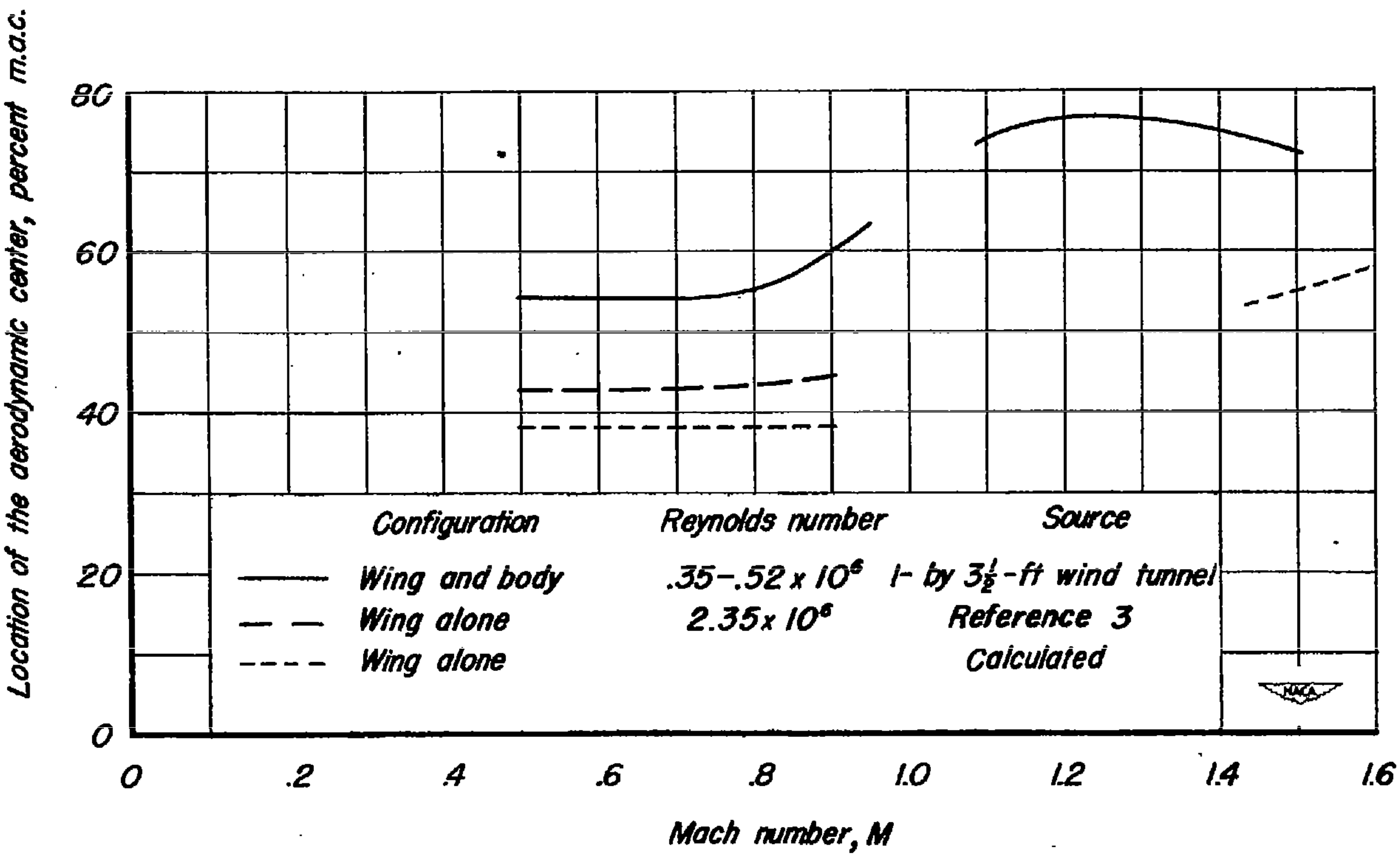

Figure 12.– Effect of Mach number on the location of the aerodynamic center.

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